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PRESSURE MEASUREMENTS IN THE NEAR FIELD OF A THREE-DIMENSIONAL DELTA WING MODEL

Ernest J. Lucas
ARO, Inc.

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March 1969

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NOTICES

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FOREWORD

The work reported herein was done at the request of the Air Force Office of Scientific Research (AFOSR), Air Force Systems Command (AFSC), for Aerospace Research Associates (ARA, Inc.), West Covina, California, under Program Element 61102F, Project 9781.

The results of tests presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), AFSC, Arnold Air Force Station, Tennessee, under Contract F40600-69-C-0001. The test program was conducted from November 6 through December 9, 1968, under ARO Project No. VT0864, and the manuscript was submitted for publication on February 4, 1969.

This technical report has been reviewed and is approved.

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ABSTRACT

Tests were conducted to determine the near field static pressure signatures created by a 70-deg sweep delta wing model. The model was tested with three leading edges of 0-, 7-, and 11-deg camber and with three leading-edge slot width conditions (0, 0.03, and 0.10 in.). Static pressure signatures were measured with a traversing disk-type pressure probe. A 9-deg half-angle cone was used to generate a flow field through which the disk static pressure probe was traversed for the basic probe calibration. The tests were conducted at free-stream Mach numbers 1.76, 2.00, and 3.00 at model angles of attack of 0 and 6 deg and Reynolds numbers, based on the 10-in. model root chord, from 1.5×10^6 to 4.2×10^6.

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NOMENCLATURE

M_∞	Free-stream Mach number
p	Probe static pressure, psia
p_∞	Free-stream static pressure, psia
Re_ℓ	Free-stream Reynolds number, based on model root chord length of 10.0 in.
Re_∞	Free-stream unit Reynolds number, in.$^{-1}$
x	Axial distance from model nose, positive downstream, in. (see Fig. 3b)
x_c	Axial distance from calibration cone nose, positive downstream, in.
z	Vertical distance from model nose, positive down, in. (see Fig. 3b)
α	Angle of attack, deg

SECTION I
INTRODUCTION

These tests constitute the second phase of a wind tunnel test program for sonic boom reduction studies on a three-dimensional delta wing model. Aerospace Research Associates (ARA, Inc.) is conducting a research program to investigate reduction of the sonic boom overpressure while maintaining or increasing the vehicle lift-to-drag ratio. The objective of the first-phase test program (Ref. 1) was to investigate the effect of various leading-edge slot widths and cambers on the vehicle lift-to-drag ratio. In the present test, the effect of leading-edge camber and slot width on the near field static pressure signatures was investigated. Three leading-edge configurations, cambered 0, 7, and 11 deg, were tested with nominal slot widths of 0, 0.03, and 0.10 in.

For the disk static pressure probe calibration, a 9-deg half-angle cone was used to generate a flow field through which the disk probe was traversed. These data were compared with theoretical results from a numerical solution of the nonlinear differential equations of Taylor and Maccoll described in Ref. 2.

The tests were conducted in the 40-in. supersonic tunnel (Gas Dynamic Wind Tunnel Supersonic (A)) of the von Kármán Gas Dynamics Facility (VKF) at Mach numbers of 1.76, 2.00, and 3.00 at angles of attack of 0 and 6 deg and Reynolds numbers, based on the wing root chord of 10 in., from 1.5×10^6 to 4.2×10^6.

SECTION II
APPARATUS

2.1 WIND TUNNEL

Tunnel A is a continuous, closed-circuit, variable density wind tunnel with an automatically driven flexible-plate-type nozzle and a 40- by 40-in. test section. The tunnel is operated at Mach numbers from 1.5 to 6 at maximum stagnation pressures from 29 to 200 psia, respectively, and stagnation temperatures up to 760°R ($M_\infty = 6$). Minimum operating pressures range from about one-tenth to one-twentieth of the maximum pressures. A more complete description of the tunnel and airflow calibration information may be found in Ref. 3.

2.2 MODEL

The model, supplied by ARA, Inc., was a 70-deg sweep delta wing with a 10-deg included thickness angle (angle between upper and lower surfaces in a longitudinal cross section) and a 10-in. centerline chord length. Photographs of the basic model configuration (zero camber leading edge, slot sealed) installed in Tunnel A with the probe system and of the model with the three leading edges are presented in Figs. 1a and b (Appendix), respectively. The three leading edges were cambered 0, 7, and 11 deg in a plane parallel to the free-stream flow (see Fig. 2). The slot between the main body and the leading edge was measured normal to the centerbody-leading-edge junction as shown in Fig. 2.

It should be noted that the slot was not uniform (maximum deviation from the nominal values was ±0.01 in.) along the leading edge. An epoxy was used to seal the slot for the closed slot configuration. Also, the leading edge was thicker than the centerbody at their junction such that a discontinuity (0.006 to 0.025 in.) existed in the model surface contour.

A 9-deg half-angle sharp cone 13.73 in. long was used as a flow field generator for the disk probe calibration. The cone was mounted in the tunnel at zero angle of attack in the same tunnel location as the delta wing (see Fig. 3). A summary of the delta wing model configurations and test conditions and cone calibration test conditions is given below.

TABLE I
TEST SUMMARY

Configuration		Angle of Attack, α, deg	$Re_\infty \times 10^{-6}$, in.$^{-1}$	M_∞
Leading-Edge Camber, deg	Slot Width, in.			
0	0	0, 6	0.15 and 0.34	1.76, 2.0, 3.0
0	0.03 and 0.10	6	0.34	↓
7	0	0, 6	0.15 and 0.34	↓
7	0.03 and 0.10	6	0.34 and 0.42	↓
11	0	0, 6	0.15 and 0.34	↓
11	0.03 and 0.10	6	0.34	1.76, 2.0, 3.0
Calibration Cone		0	0.34	1.76, 2.0, 3.0

2.3 SURVEY PROBE INSTRUMENTATION AND TECHNIQUE

The static pressure probe used for these surveys was a 1-in.-diam disk (Fig. 3a). The probe was oriented such that the probe face was aligned with and parallel to the model vertical centerline plane, which was coincident with the wind tunnel x-z plane (see Fig. 3b). With this orientation, and traversing axially along the model centerline plane only, the probe surface was always in the plane of the flow direction. To reduce the effect of variations in free-stream static pressure created by small fluctuations in the tunnel supply pressure, a test section sidewall static pressure measurement was used to electrically cancel these effects, pressure measurement was used to electrically cancel these effects, as suggested by Ref. 4. No effort was made to correct the pressure signatures for the test section axial static pressure gradient (maximum $\Delta p_\infty / p_\infty = \pm 3$ percent over $\Delta x = 53$ in.). The probe system provided a remotely controlled longitudinal traverse of 53 in.

Miniature transducers used to measure the probe static pressures were developed at VKF primarily for use in the VKF hotshot tunnels. The transducers were calibrated for 3.0-psid full scale, and a variable reference pressure was utilized to stay within this limit. The transducers are considered accurate to within ±2 percent of the calibrated range.

The data were recorded on analog (x-y) plotters, which provided a continuous trace of the probe static pressure with axial position (x), and on tape for computer computation of discrete points for tabulated data.

SECTION III
RESULTS AND DISCUSSION

3.1 PROBE CALIBRATION

Results of the probe calibration in a conical flow field are compared in Fig. 4 with an inviscid theoretical prediction (using Ref. 2) for the 9-deg half-angle cone. The agreement between the measured pressure rise across the shock and the inviscid calculation is observed to be good at Mach numbers 2.0 and 3.0, although the flow field pressures behind the shock were lower than theory. The measured pressures were consistently lower than theoretical at Mach number 1.76.

The disk probe was traversed ahead of the conical calibration body and the delta wing model leading-edge shocks. The probe's ability to measure free-stream static pressure was thereby determined and is shown in the data traces (Figs. 4 to 7). The maximum deviation from the calculated free-stream static pressure, which is based on an average test section Mach number from tunnel calibration, was ±5 percent. A portion of this deviation can also be attributed to test section axial free-stream pressure gradients and to small misalignments of the probe relative to the free-stream flow direction.

3.2 DELTA WING SURVEYS

The leading-edge slots were expected to be effective only at angle of attack, so only the slot-sealed configurations were tested at $\alpha = 0$. The static pressure signatures for the three leading-edge cambers are presented in Fig. 5 for the three Mach numbers tested.

Data were obtained at $\alpha = 6$ deg for the three leading-edge cambers at slot widths of 0, 0.03, and 0.10 in., and the results are presented in Figs. 6a, b, and c, respectively. These data have also been plotted to show the effect of varying the slot width for the three wing leading edges in Figs. 7a, b, and c. There was no apparent reduction in the shock pressure rise with camber, but the maximum slot width (0.10-in.) changed the shape of the pressure signature at the z = 5-in. station in Figs. 7a and c (dashed curve). This was apparently a local disturbance, and the effect on the far field signature may be negligible since the effect was less prominent at the z = 17-in. station.

It should be noted that because of the model size and limited downstream probe travel, the complete typical "N"-type static pressure signatures could not be obtained at $M_\infty = 3.0$. The irregularities in the "plateau" region of the three-dimensional flow field signatures are attributed to the model slot or the step which existed when the slot was sealed. The final analysis of the data with respect to sonic boom reduction will be accomplished by ARA, Inc.

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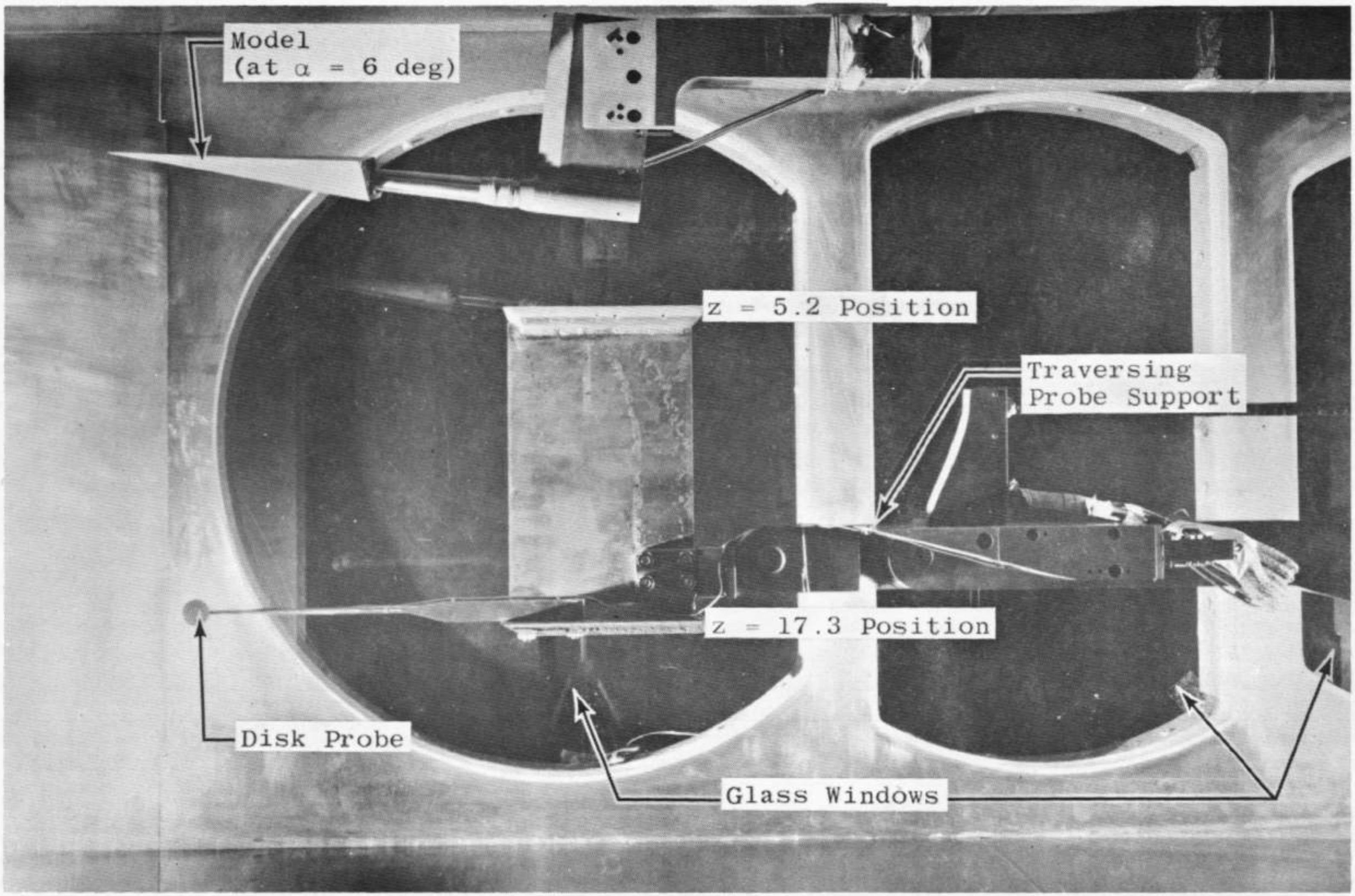
3. Test Facilities Handbook (Seventh Edition). "von Kármán Gas Dynamics Facility, Vol. 4." Arnold Engineering Development Center, July 1968.

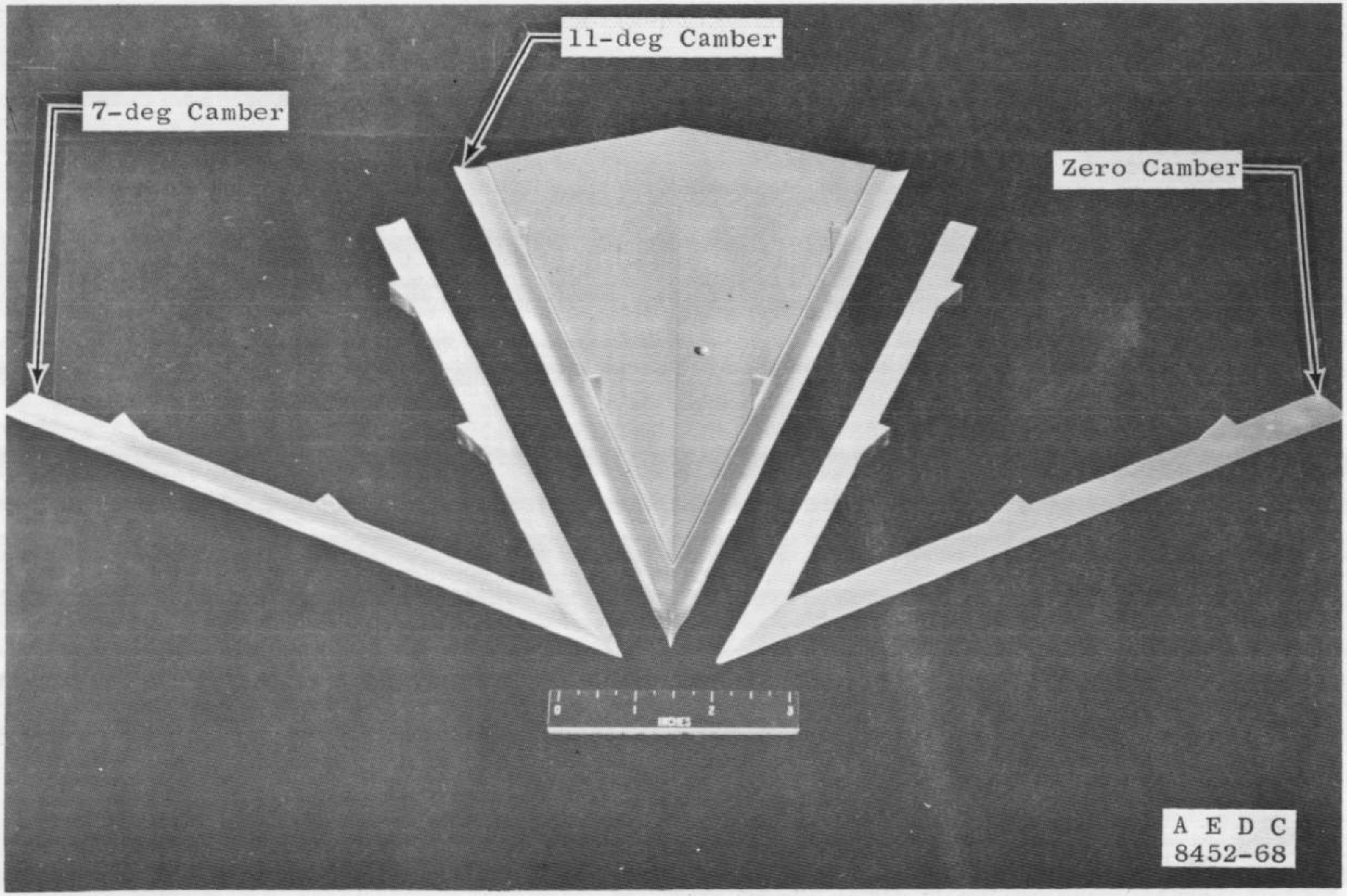
4. Pate, S. R. "Pressure Measurements in the Near Field of a Supersonic Two-Dimensional Wing-Jet Configuration." AEDC-TR-68-131 (AD834772), June 1968.

APPENDIX
ILLUSTRATIONS



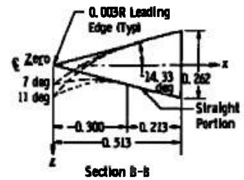
a. Basic Configuration (Zero Camber, Slot Sealed) Installed in Tunnel A

Fig. 1 Model Photographs



b. Centerbody with Leading Edges

Fig. 1 Concluded

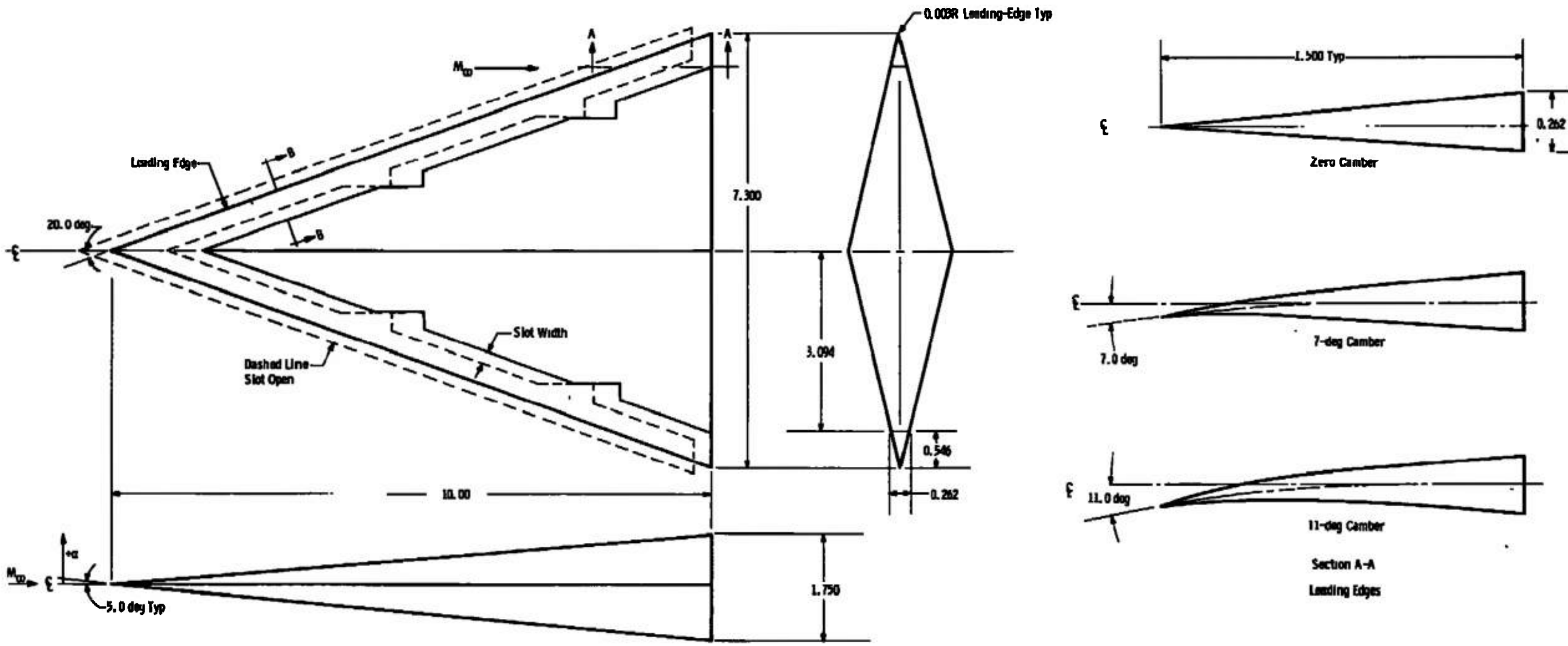


Perpendicular Section through Wing Leading Edge

Contour Coordinates of the Leading Edges				
Camber	7 deg		11 deg	
x	z_{upper}	z_{lower}	z_{upper}	z_{lower}
0	+0.080	+0.080	+0.130	+0.130
0.05	+0.043	+0.068	+0.078	+0.103
0.10	+0.010	+0.061	+0.032	+0.083
0.15	-0.018	+0.058	-0.006	+0.071
0.20	-0.042	+0.060	-0.037	+0.066
0.25	-0.062	+0.066	-0.060	+0.067
0.30	-0.077	+0.077	-0.077	+0.077
**				
0.513	-0.131	+0.131	-0.131	+0.131

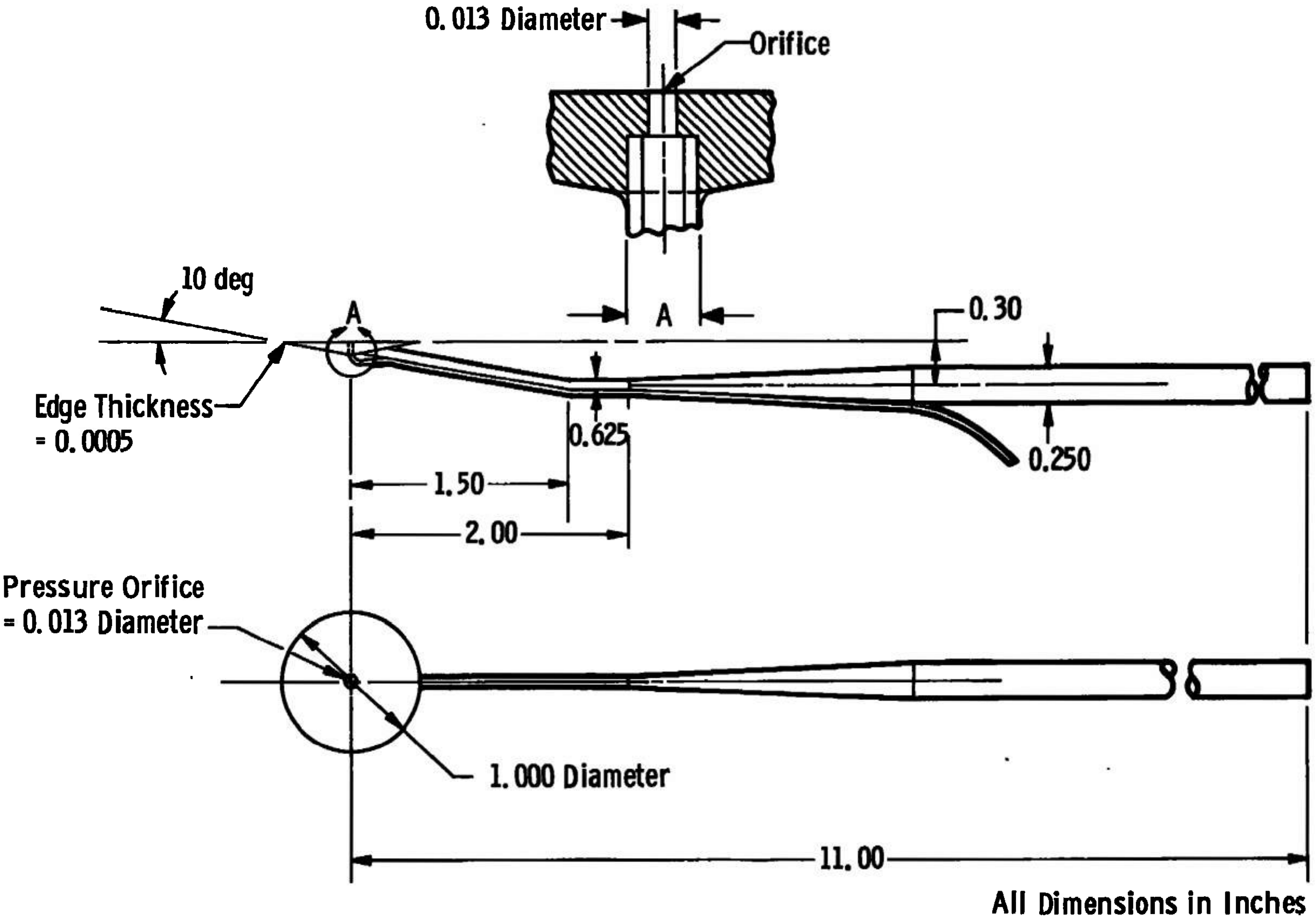
Zero camber leading edge is straight with half-angle 14.33 deg.

**Straight line joins station x = 0.30 and 0.513.



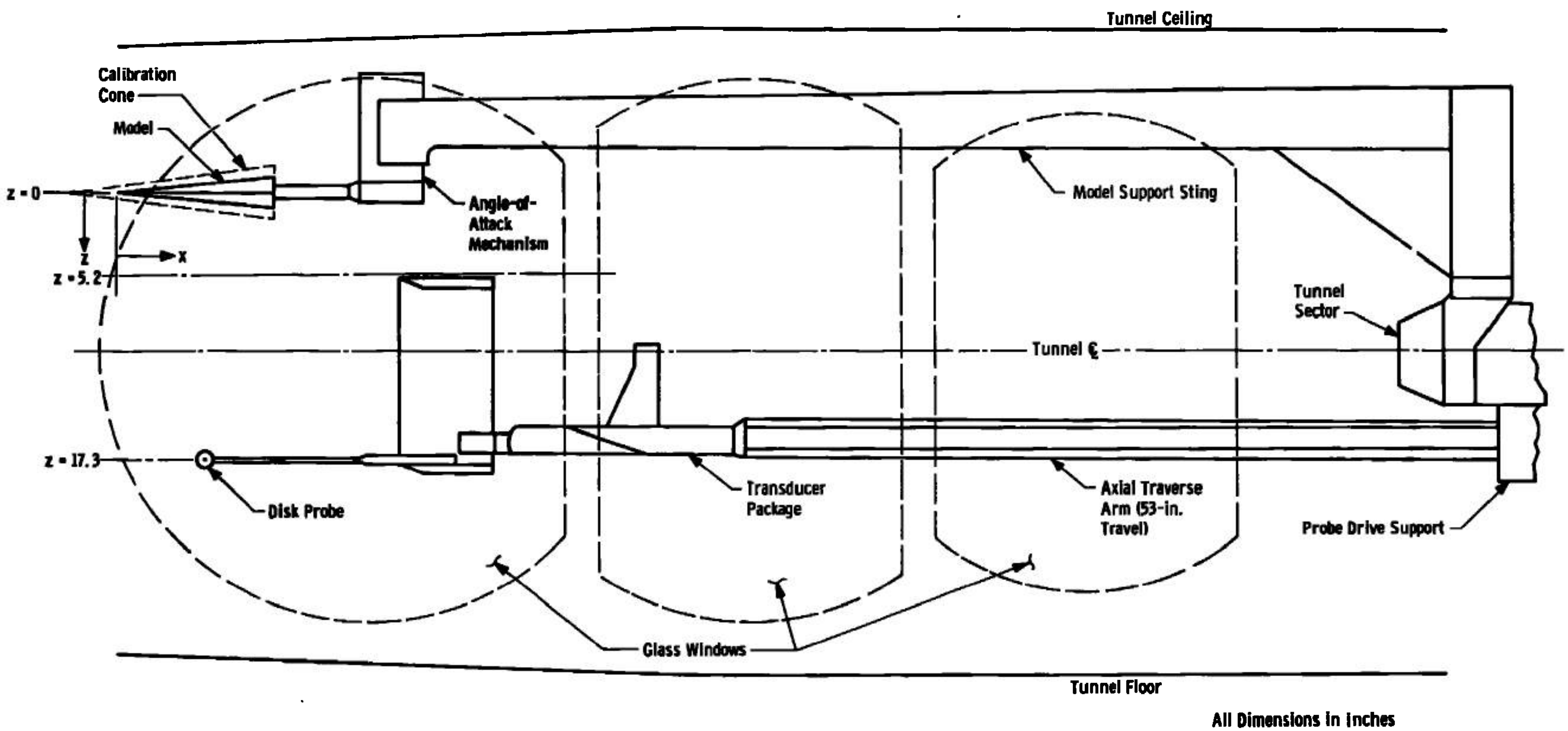
All Dimensions in Inches

Fig. 2 Model Geometry



a. Disk Probe Details

Fig. 3 Probe and Survey System



b. Survey System Setup

Fig. 3 Concluded

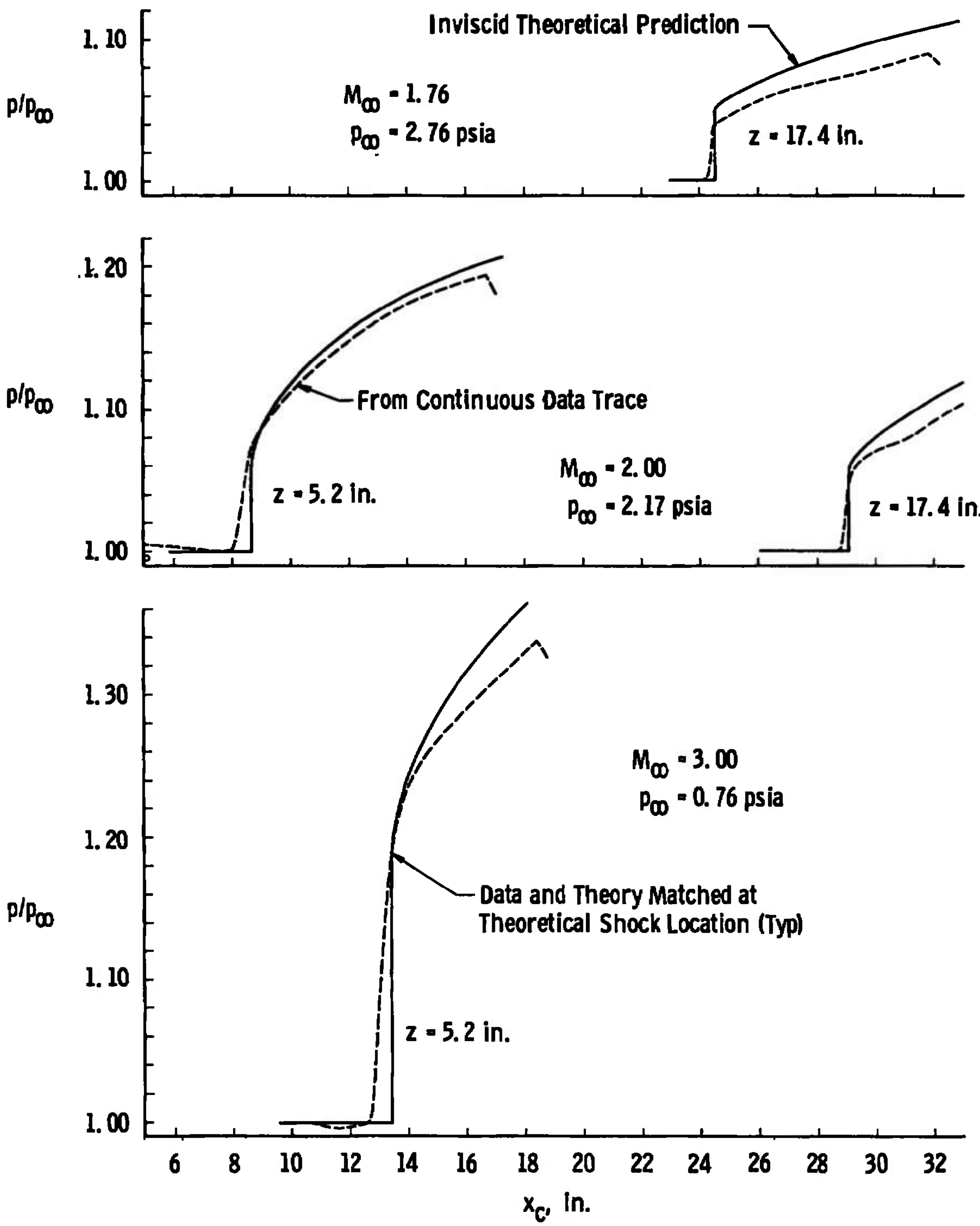


Fig. 4 Probe Static Pressure Compared to Conical Flow Theory

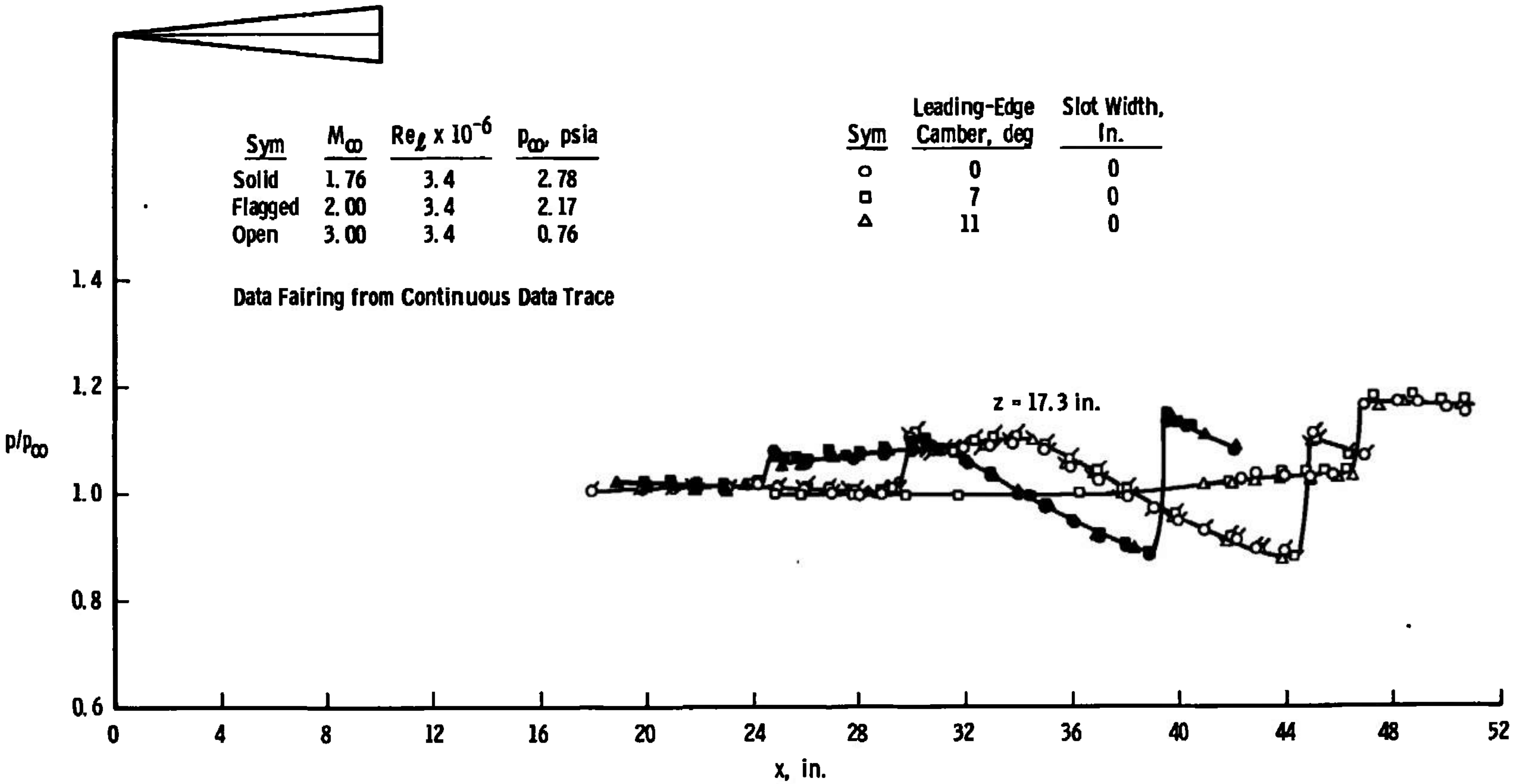
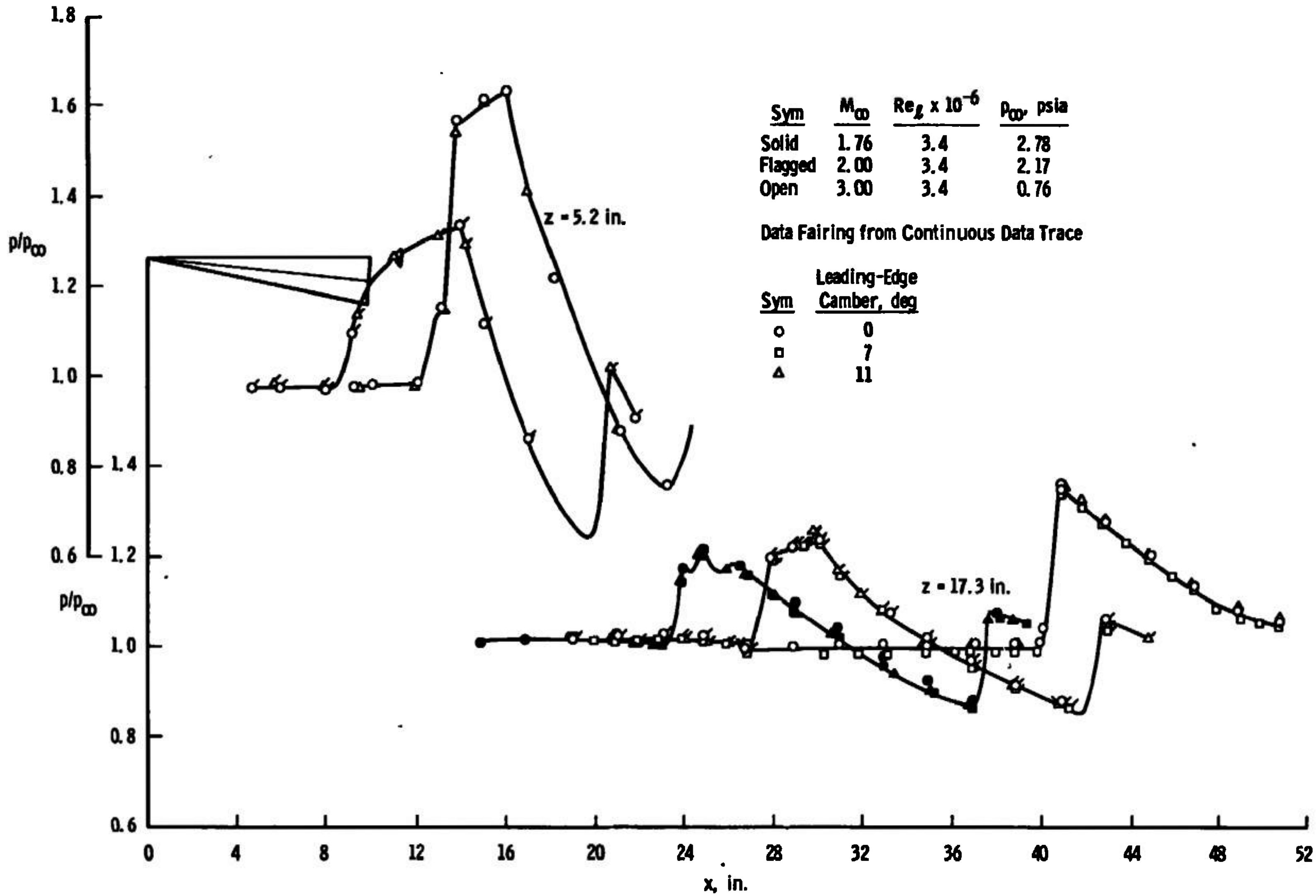
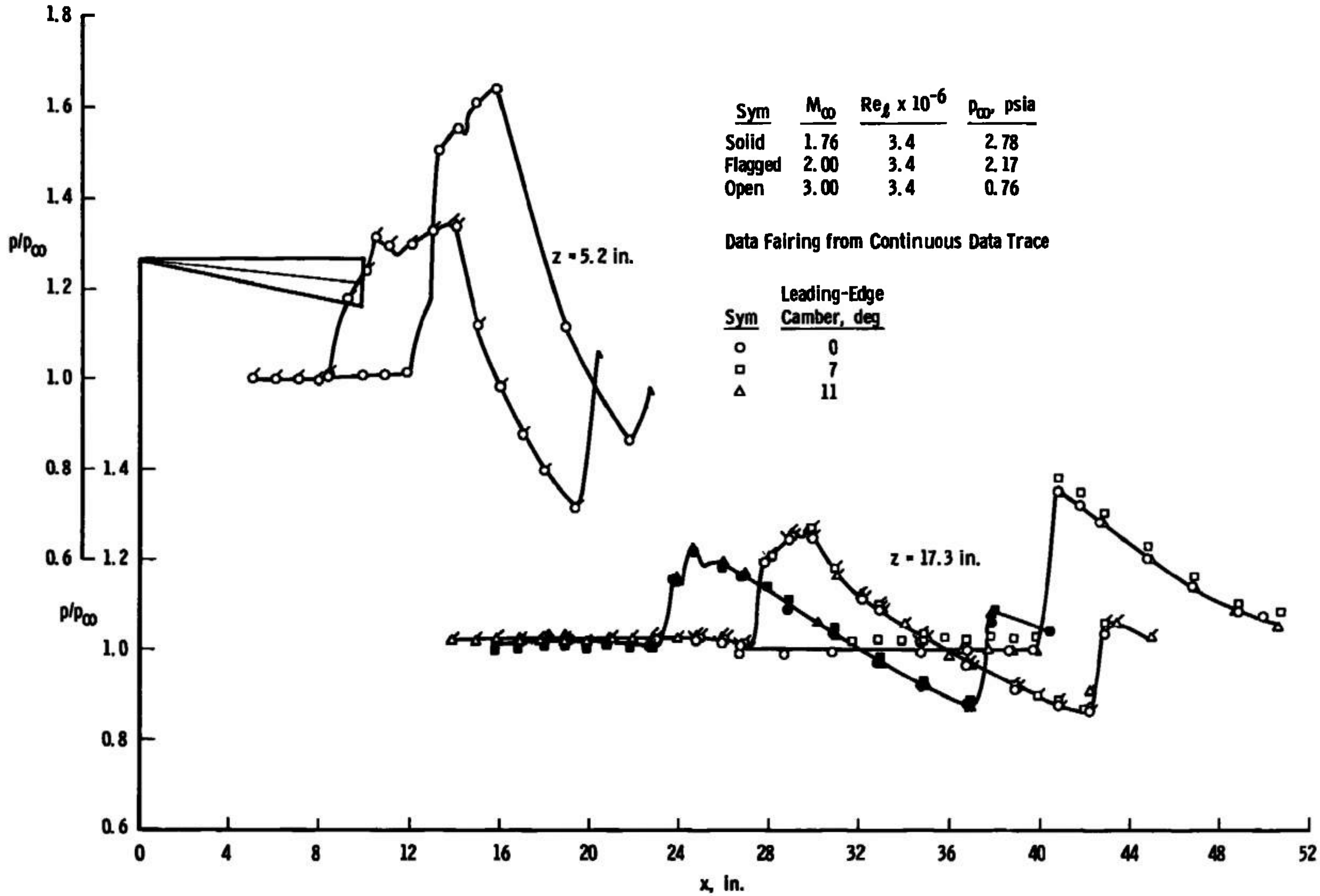


Fig. 5 Leading-Edge Effect for $\alpha = 0$



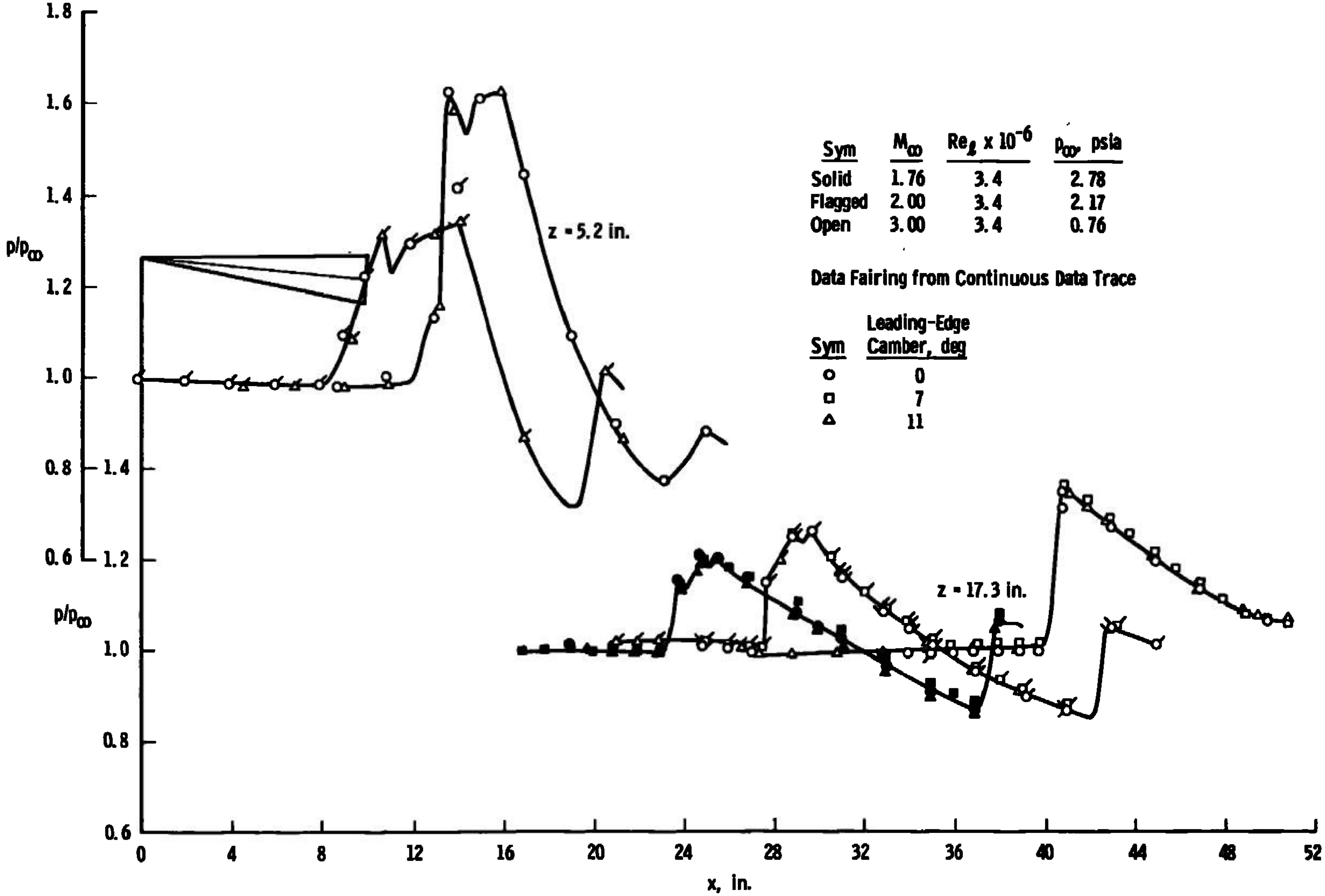
a. Slot Sealed

Fig. 6 Leading-Edge Effect for $\alpha = 6$ deg



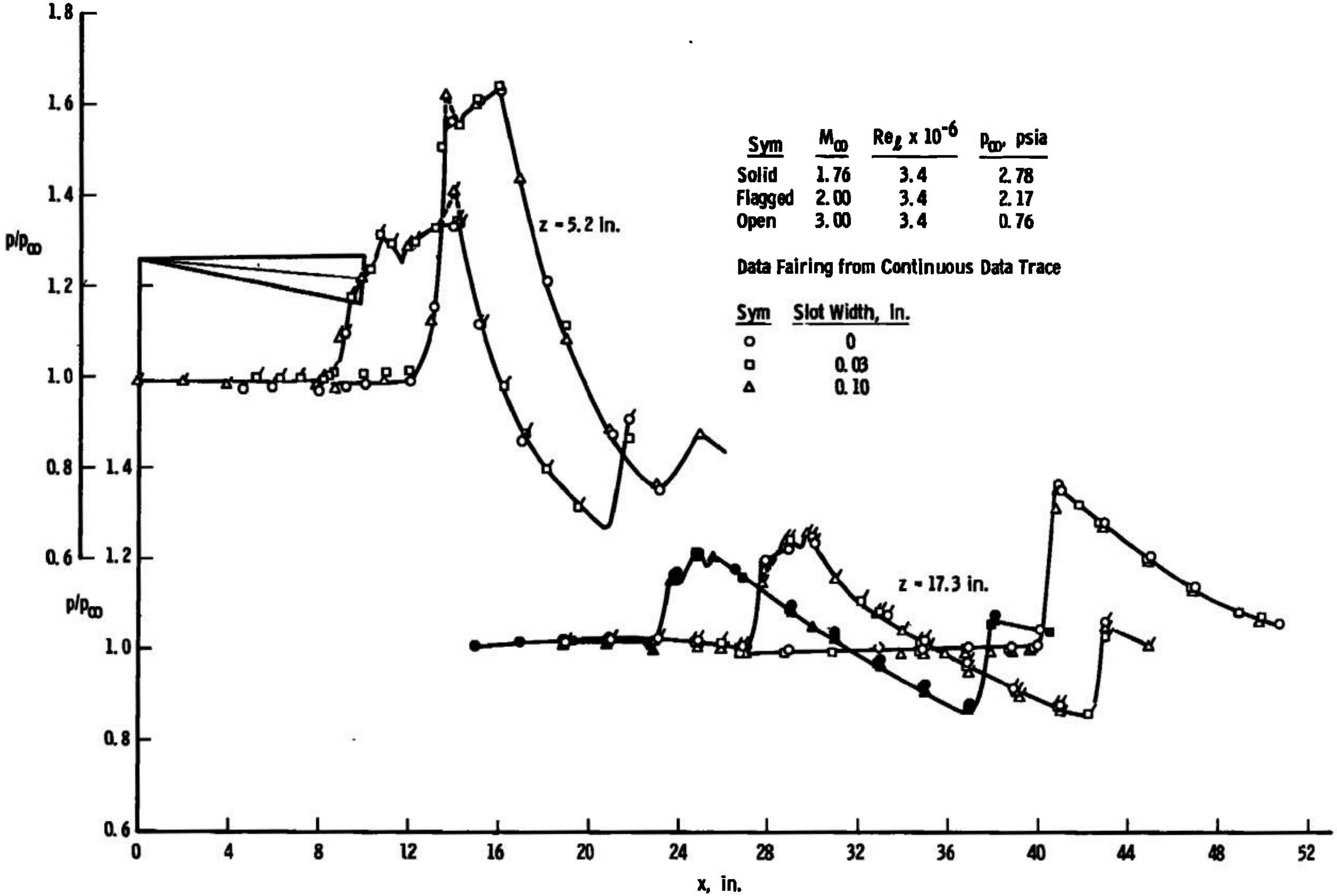
b. 0.03-in. Slot

Fig. 6 Continued



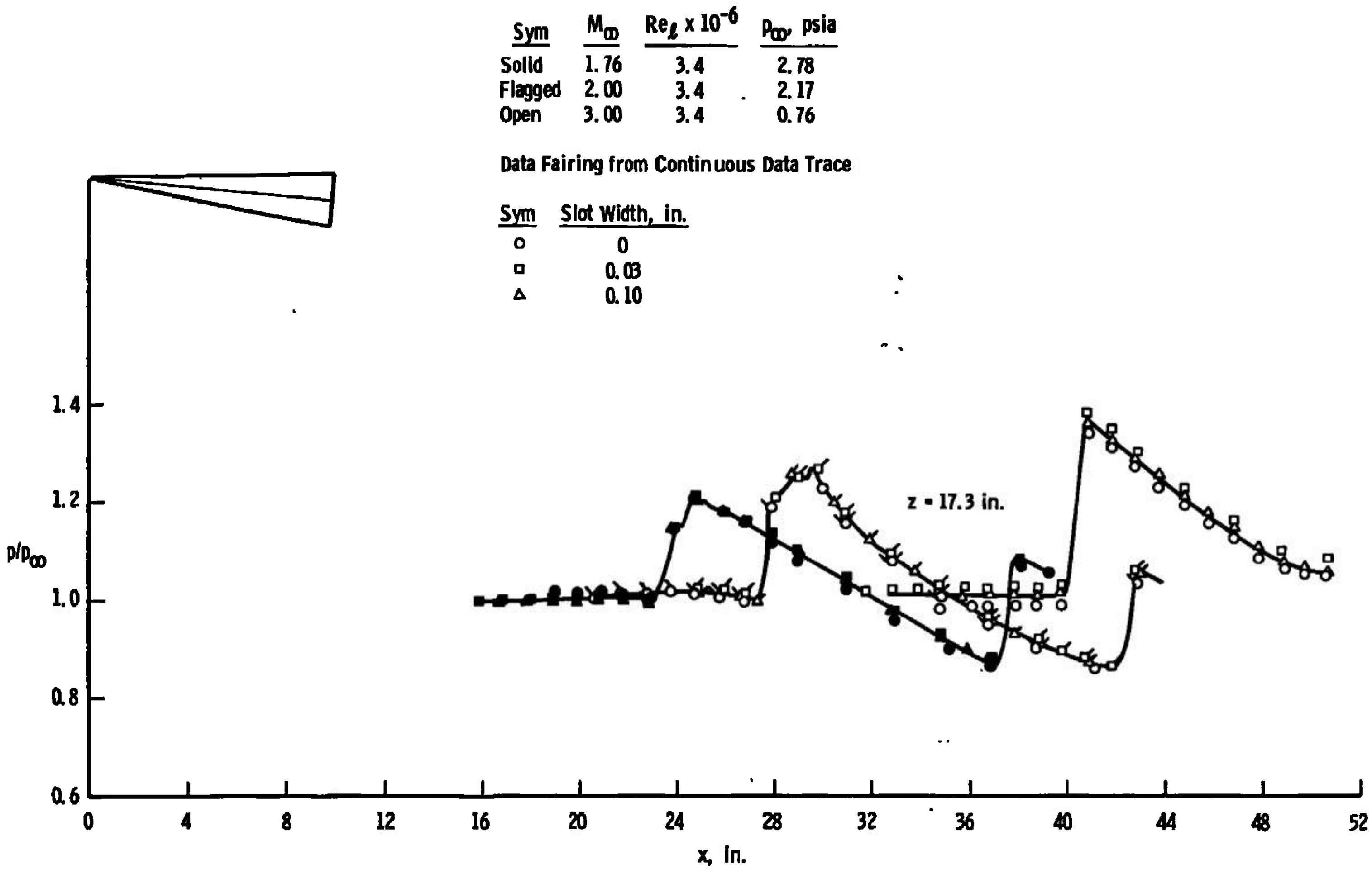
c. 0.10-in. Slot

Fig. 6 Concluded



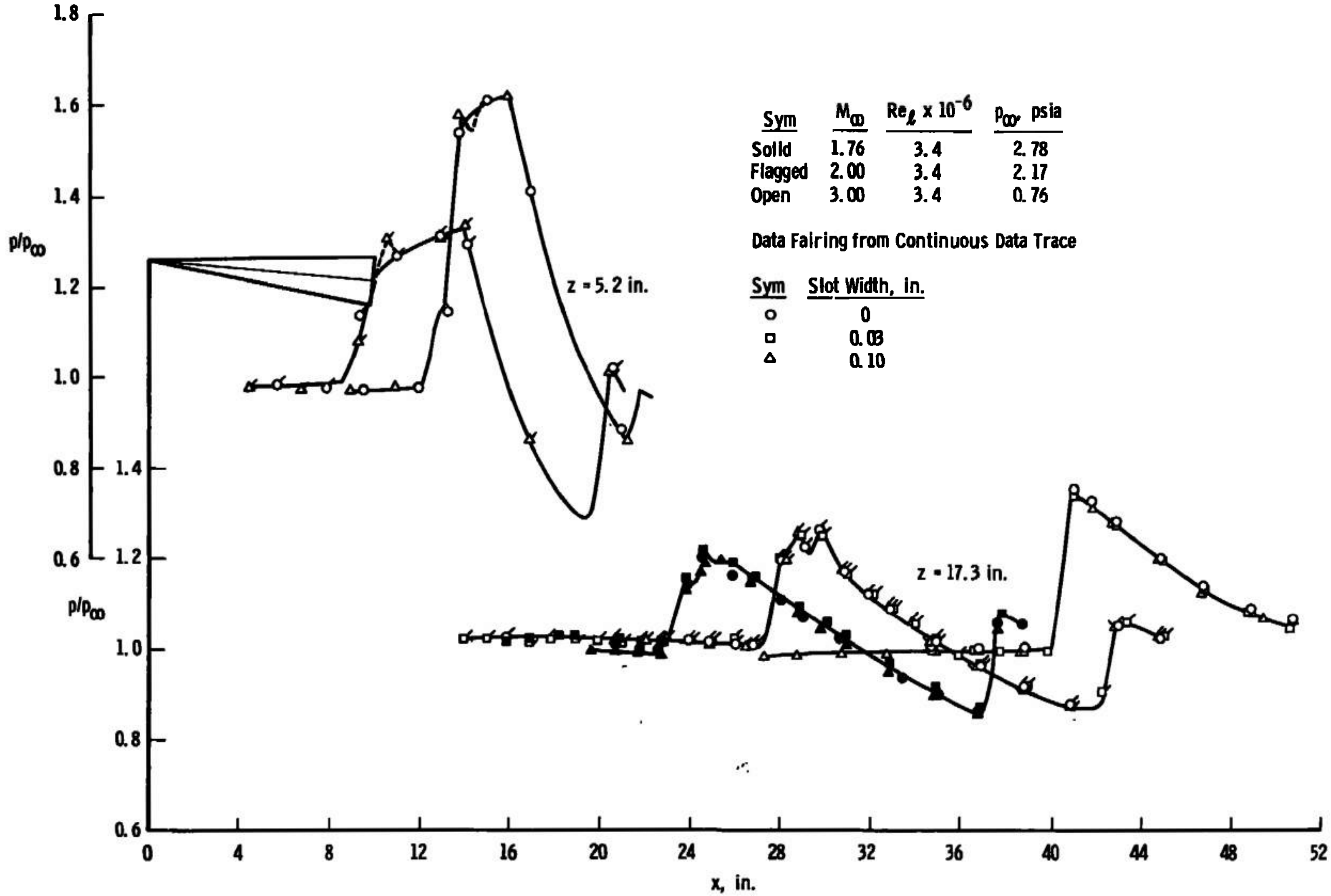
a. 0 Camber

Fig. 7 Slot Effects at $\alpha = 6$ deg



b. 7-deg Camber

Fig. 7 Continued



c. 11-deg Camber

Fig. 7 Concluded

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wind tunnel models						
static pressure						
pressure signatures						
aircraft noise						
aerodynamic configurations						
supersonic flow						

1. Wings -- Pressure distribution
2. Triangular " -- " "
3. Swept wings -- " "

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